TURBOFAN AIRCRAFT ENGINE

[0001] The present invention relates to a turbofan aircraft engine having a primary duct including a combustion chamber, a first turbine disposed downstream of the combustion chamber, a compressor disposed upstream of the combustion chamber and coupled to the first turbine, and a second turbine which has a plurality of turbine stages having rotor blades and is disposed downstream of the first turbine and coupled via a speed reduction mechanism to a fan for feeding a secondary duct. The invention further relates to a passenger jet for at least 10 passengers which has a turbofan aircraft engine of this type, as well as to a method for designing such a turbofan aircraft engine.

[0002] Today, most engines of modern passenger jets are turbofan aircraft engines. In order to increase the efficiency thereof and/or to reduce noise emission, so-called "geared turbofans" are known from in-house practice. In such geared turbofans, the fan and the turbine driving it are coupled via a speed reduction mechanism.

[0003] This provides new degrees of freedom in the design of the engine components.

SUMMARY OF THE INVENTION

[0004] It is an object of an embodiment of the present invention to provide an improved passenger jet.

[0005] The present invention provides a turbofan aircraft engine that has a primary gas duct (hereinafter also referred to as "primary duct") for a so-called "core flow." The primary duct includes a combustion chamber, in which, in an embodiment, air that is drawn-in and compressed is burned together with supplied fuel during normal operation. The primary duct includes a first turbine which is located downstream, in particular immediately downstream, of the combustion chamber and which, without limiting generality, is hereinafter also referred to as "high-pressure turbine". The axial location information "downstream" refers in particular to a through-flow during, in particular, steady-state operation and/or normal operation. The first turbine or high-pressure turbine may have one or more turbine stages, each including a rotor blade array and preferably a stator vane array downstream or upstream thereof, and is coupled, in particular fixedly connected, to a compressor of the primary duct such that they rotate at the same speed. The compressor is preferably disposed immediately upstream of the combustion change and, without limiting generality, is hereinafter also referred to as "high-pressure compressor". The high-pressure compressor may have one or more stages, each including a rotor blade array and preferably a stator vane array downstream or upstream thereof. The high-pressure compressor, combustion chamber and high-pressure turbine together form a so-called "core engine."

[0006] The turbofan aircraft engine has a secondary duct, which is preferably arranged fluidically parallel to and/or concentric with the primary duct. A fan is disposed upstream of the secondary duct to draw in air and feed it into the secondary duct. The fan may have one or more axially spaced-apart rotor blade arrays; i.e., rows of rotor blades distributed, in particular equidistantly distributed, around the circumference thereof. A stator vane array may be disposed upstream and/or downstream of each rotor blade array of the fan. In one embodiment, the fan is an upstream-most or first or forwardmost rotor blade array of the engine, while in another embodiment, the fan is a downstream-most or last or rearwardmost rotor blade array of the engine ("aft fan"). In one embodiment, the fan is adapted or designed to feed also the primary duct and/or is preferably disposed immediately upstream of the primary duct and/or the secondary duct. At least one additional compressor may be disposed between the fan and the first compressor or high-pressure compressor. Without limiting generality, the additional compressor is also referred hereinafter to as "low-pressure compressor."

[0007] The fan is coupled via a speed reduction mechanism to a second turbine of the primary duct. The second turbine is disposed downstream of the high-pressure turbine and, without limiting generality, is hereinafter also referred to as "low-pressure turbine". The second turbine or low-pressure turbine has a plurality of turbine stages, each including a rotor blade array including a plurality of circumferentially distributed rotor blades and, in an embodiment, a stator vane array which includes a plurality of circumferentially distributed stator vanes and is disposed upstream or downstream of the rotor blade array. In one embodiment, at least one additional turbine may be disposed between the high-pressure and low-pressure turbines and, in one embodiment, several or all turbine stages coupled to the fan via the speed reduction mechanism together form the second turbine or low-pressure turbine according to the present invention. In one embodiment, the fan and the low-pressure turbine may be coupled via a low-pressure shaft extending through a concentric hollow shaft that couples the high-pressure compressor and the high-pressure turbine. The speed reduction mechanism may include a transmission, in particular, a single- or multi-stage gear drive. In one embodiment, the speed reduction mechanism may have an in particular fixed speed reduction ratio of at least 2:1, in particular at least 3:1, and/or no greater than 11:1, in particular no greater than 4:1, between a rotational speed of the low-pressure turbine and a rotational speed of the fan. As used herein, a speed reduction mechanism is understood to mean, in particular, a non-rotatable coupling which converts a rotational speed of the low-pressure turbine to a lower rotational speed of the fan.

[0008] The number of all turbine stages of the second turbine, in particular of all axially spaced-apart rotor blade arrays that are coupled to the fan via the speed reduction mechanism, defines a total stage count of all turbine stages of the second turbine. The number of all rotor blades and stator vanes of all turbine stages of the second turbine together defines a total blade count of all rotor blades and stator vanes of the second turbine.

[0009] At a predetermined design point, each turbine stage of the second turbine has a (design) stage pressure ratio of the (design) pressure at the inlet to the pressure at the exit of this turbine stage. At the predetermined design point, the second turbine as a whole has a (design) total pressure ratio of the (design) pressure at the inlet of the upstreammost or first turbine stage to the (design) pressure at the exit of the downstreammost or last turbine stage of the second turbine. This (design) total pressure ratio is, in particular, equal to the product of the stage pressure ratios of all turbine stages of the second turbine.

[0010] The predetermined design point may in particular be an operating point of the turbofan aircraft engine which, in an embodiment, may be defined by a predetermined rotational speed and/or a predetermined mass flow of air through the turbofan aircraft engine and which may in particular be the so-called "redline point"; i.e., an operating point of maximum